



RESEARCH MEMORANDUM

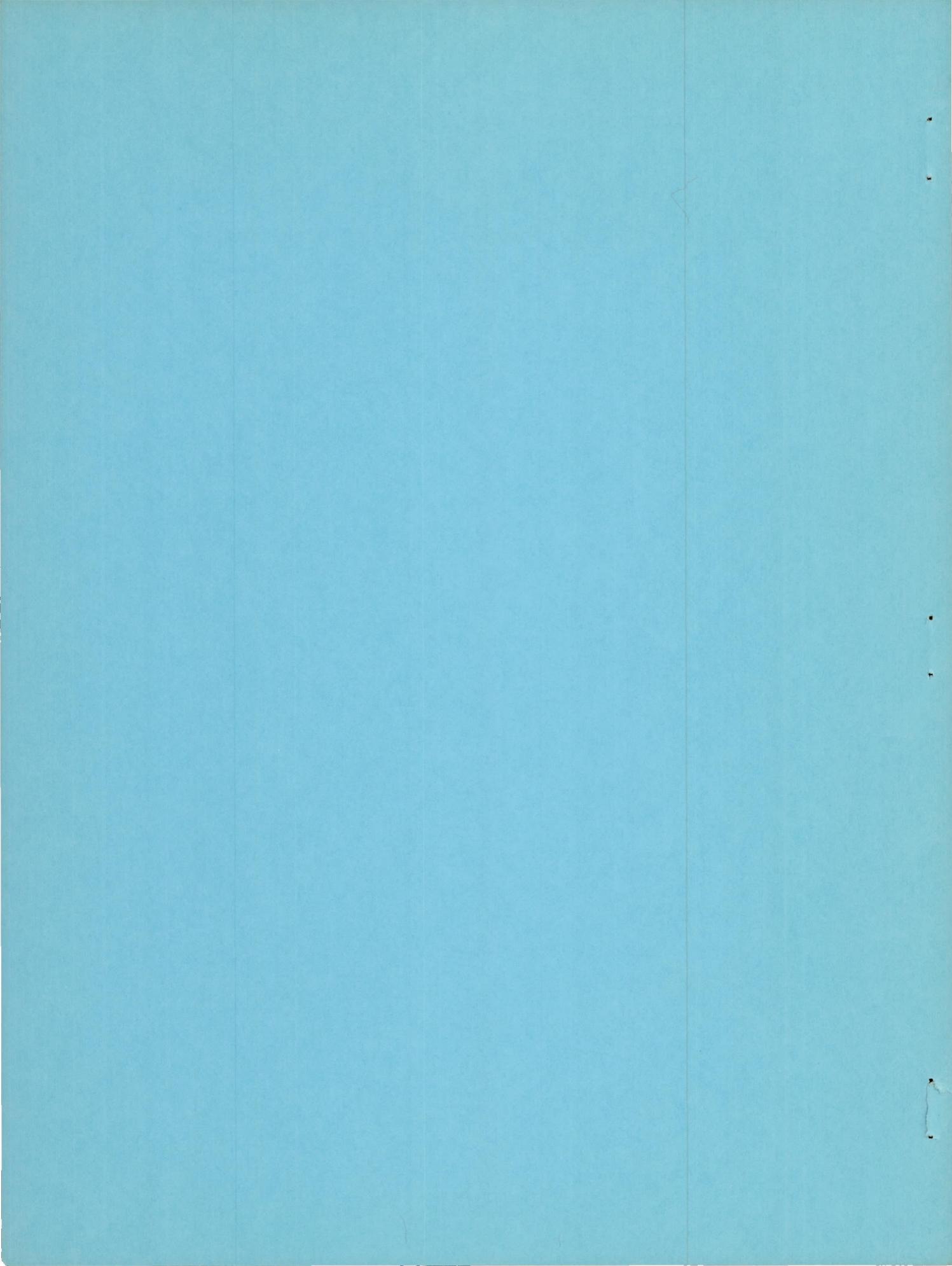
A STUDY OF THE USE OF FREON-12 AS A WIND-TUNNEL TESTING
MEDIUM AT LOW SUPERSONIC MACH NUMBERS

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

A comparison between the force data obtained on a wing in air and Freon-12 at a low supersonic Mach number has been obtained from the pressure coefficients measured at three spanwise stations on a 45° sweptback wing at angles of attack of 0°, 2°, 4°, and 6°. The pressure-coefficient data obtained in Freon-12 were converted to equivalent air values by the method presented in NACA RM L51III and very close agreement was obtained between these converted data and the data actually obtained in air. Empirical conversion factors to be applied to data obtained through the transonic Mach number range in Freon-12 for reduction to equivalent air values are presented as computed on the assumption of streamline similarity of the flow in both Freon-12 and air.

INTRODUCTION

At the present time Freon-12 is used as a wind-tunnel testing medium at high subsonic Mach numbers in the Langley low-turbulence pressure tunnel (ref. 1). The use of Freon-12 in the place of air has permitted an increase in the wind-tunnel test Mach number from 0.4 to approximately 1.0 without any additional power requirements. References 2 and 3 have indicated the adaptability of Freon-12 as a wind-tunnel testing medium; however, the utility of data obtained in tests in an atmosphere of Freon-12 is contingent upon the correspondence between data so obtained and data obtained in similar tests in air. It has been found in tests at subsonic Mach numbers with Freon-12 that any differences between the results of flow tests in air and Freon-12 are small and, further, that these small differences can largely be resolved by the application of a conversion method to the data which brings the results into substantial agreement. The relation utilized in the subsonic Mach number range to effect this conversion of the data is based upon the concept of geometrical similarity of the flow patterns in the two testing mediums. The conversion method is described and some theoretical justification for its application is offered in reference 1 on the basis of the transonic similarity rule developed in reference 4.

The ability to convert data obtained in Freon-12 to equivalent air data in the subsonic Mach number range under widely varying flow conditions, which included flows with shocks at various chordwise positions of the models tested, suggested the use of Freon-12 at low supersonic Mach numbers. Because of the greater divergence between the flow characteristics in air and Freon-12 at supersonic Mach numbers, it was not apparent that the same conversion methods based on the concept of geometrical flow similarity would be applicable even at low supersonic free-stream Mach numbers. The use of Freon as a testing medium at low supersonic Mach numbers was accordingly investigated by temporary modification of the test section of the Langley low-turbulence pressure tunnel to permit operation with Freon-12 at a Mach number of approximately 1.185.

This report compares pressure-distribution measurements on a 45° sweptback wing in supersonic flow in Freon-12 with similar measurements previously made on the same wing in air in the Langley 8-foot high-speed tunnel. Small differences, which were again found to exist between the results obtained in the two testing mediums due to the difference in specific-heat ratio γ between the two gases, are largely resolved by application of the same conversion method developed for subsonic speeds in reference 1. The method proposed for the conversion of force and moment coefficients measured in Freon-12 to their equivalent air values at subsonic Mach numbers is described in reference 1 and the same techniques have been applied herein to extend these force conversion factors through a range of low supersonic Mach numbers.

SYMBOLS

A	stream-tube area
b	span of wing
c	section chord of wing, measured parallel to plane of symmetry of model
x	distance from leading edge along section chord of wing
H	total pressure
M	Mach number
ρ	mass density
V	velocity
q	dynamic pressure, $\frac{1}{2}\rho V^2$

p	static pressure
P	pressure coefficient, $\frac{p - p_0}{q_0}$
P_R	resultant pressure coefficient, $P_L - P_U$
y	distance normal to free stream in lift direction
α	angle of attack
γ	ratio of specific heat at constant pressure to specific heat at constant volume
c_d	wing-section wake drag coefficient
C_{DL}	wing-drag coefficient due to lift
C_N	wing normal-force coefficient
c_m	section pitching-moment coefficient about 0.25c
c_n	section normal-force coefficient
Δc_d	coefficient in Freon-12 minus corresponding coefficient in air
ΔC_N	
Δc_m	
Δc_n	
ΔP_R	

Subscripts:

A	air
F	Freon-12
U	upper surface
L	lower surface
o	conditions in free stream

l conditions in wake
 cr conditions at a local Mach number of 1.0
 l local conditions

METHODS FOR CONVERTING FREON-12 DATA TO AIR DATA

The conversion of Freon-12 pressure coefficients to equivalent air pressure coefficients is based upon the concept of geometrical similarity of the streamline flow patterns in the two testing mediums. With this concept the stream-tube area ratio A_{cr}/A for any position of the flow in Freon-12 is considered equal to the stream-tube area ratio for the same position of the flow in air. The variation of stream-tube area ratio with Mach number for both gases is shown in figure 1 as determined from the relation

$$\frac{A_{cr}}{A} = M \left[\frac{1 + \frac{\gamma - 1}{2} M^2}{\frac{\gamma + 1}{2}} \right]^{-\frac{\gamma + 1}{2(\gamma - 1)}} \quad (1)$$

with the values of $\gamma = 1.4$ for air and $\gamma = 1.128$ for Freon-12. The stream-tube area is a minimum in both gases where the Mach number of the flow is 1.0. This minimum value of the stream-tube area A_{cr} is taken as the basis of the relation depicted in figure 1.

The conversion of values of the pressure coefficient obtained in either medium to equivalent values in the other involves the application of the area ratio curves of figure 1 and the following relation:

$$P = \frac{2}{\gamma M_0^2} \left[\frac{\left(1 + \frac{\gamma - 1}{2} M_0^2 \right)}{\left(1 + \frac{\gamma - 1}{2} M_l^2 \right)} \right]^{\frac{\gamma}{\gamma - 1}} - 1 \quad (2)$$

For example, to convert known pressure coefficients at a given free-stream Mach number in Freon-12 to air pressure coefficients, the local Mach numbers corresponding to the known pressure coefficients in Freon-12 are determined by means of equation (2). The free-stream and local Mach numbers in Freon-12 are then converted to air Mach numbers for the same area ratio as determined by figure 1. Equation (2) is again employed with the converted air Mach numbers and the value of γ for air to

calculate the air pressure coefficients. The same procedure can be followed to convert air pressure coefficients to equivalent Freon pressure coefficients.

The above-outlined pressure-coefficient conversion method was utilized in reference 1 on a large quantity of available air pressure-coefficient data obtained for two airfoils at various angles of attack to derive corresponding Freon pressure-coefficient distributions and to compare the results of chordwise integrations of the two sets of pressure-coefficient distributions. In this manner, factors were derived to be applied to normal-force, pitching-moment, and hinge-moment coefficients obtained in Freon-12 in order to convert these coefficients directly to equivalent air coefficients through the subsonic Mach number range. An explanation was presented in reference 1 for the apparently minor effects of such variables as body shape and lifting conditions on these conversion factors which were found to be primarily dependent upon the free-stream conditions. An analytical derivation of the normal-force conversion factors will now be presented in order to provide an explanation of the order of magnitude and trends of the normal-force and moment conversion factors with free-stream Mach number.

The analytical derivation of the conversion factors was made by means of a linearization of the variation of pressure ratios with Mach number for small local loadings. The analysis is based on the concept of similar area ratios A_{cr}/A in Freon-12 and in air as are the conversion curves of reference 1.

The ratio c_{nA}/c_{nF} can be expressed as

$$\frac{c_{nA}}{c_{nF}} = \frac{\int_0^1 \left(\frac{p_L - p_U}{q_o} \right)_A d\left(\frac{x}{c}\right)}{\int_0^1 \left(\frac{p_L - p_U}{q_o} \right)_F d\left(\frac{x}{c}\right)}$$

$$= \frac{\left(\frac{q_o}{H} \right)_F \int_0^1 \left[\left(\frac{p_L}{H} \right)_A - \left(\frac{p_U}{H} \right)_A \right] d\left(\frac{x}{c}\right)}{\left(\frac{q_o}{H} \right)_A \int_0^1 \left[\left(\frac{p_L}{H} \right)_F - \left(\frac{p_U}{H} \right)_F \right] d\left(\frac{x}{c}\right)} \quad (3)$$

Values of $\frac{(q_o/H)_F}{(q_o/H)_A}$ were calculated for the same value of free-stream area ratio in air and in Freon and are presented in figure 2 (curve 1)

as a function of the Freon free-stream Mach number. Values of the integrals in equation (3) are not obtainable in so straightforward a manner. Figure 3 presents the pressure ratios p/H in air and in Freon calculated for the same values of area ratio, as a function of Freon Mach number. If the problem is linearized for small loads to the extent that the difference in p/H between the upper and lower surfaces of an airfoil at any chordwise position can be defined as the product of the rate of variation of (p/H) with free-stream Mach number and the difference in upper and lower surface local Mach numbers, then

$$\frac{\int_0^1 \left[\left(\frac{p_L}{H} \right)_A - \left(\frac{p_U}{H} \right)_A \right] d\left(\frac{x}{c}\right)}{\int_0^1 \left[\left(\frac{p_L}{H} \right)_F - \left(\frac{p_U}{H} \right)_F \right] d\left(\frac{x}{c}\right)} = \frac{\int_0^1 \frac{d\left(\frac{p}{H}\right)_A}{dM_{OF}} (\Delta M_l)_F d\left(\frac{x}{c}\right)}{\int_0^1 \frac{d\left(\frac{p}{H}\right)_F}{dM_{OF}} (\Delta M_l)_F d\left(\frac{x}{c}\right)} \quad (4)$$

or

$$\frac{\frac{d\left(\frac{p}{H}\right)_A}{dM_{OF}} \int_0^1 (\Delta M_l)_F d\left(\frac{x}{c}\right)}{\frac{d\left(\frac{p}{H}\right)_F}{dM_{OF}} \int_0^1 (\Delta M_l)_F d\left(\frac{x}{c}\right)} = \frac{d\left(\frac{p}{H}\right)_A}{d\left(\frac{p}{H}\right)_F} \quad (5)$$

In actuality, the local Mach numbers on an airfoil will range from zero to some value greater than the free-stream Mach number; however, because the linear approximation to the curves of figure 3 is actually seen to be well applicable over a wide range of local Mach numbers for a large

portion of the Mach number range, the value of $\frac{d(p/H)}{dM_F}$ for the free-

stream Mach number should closely represent the mean value of this parameter for all the local Mach numbers on the airfoil. Equation (5) is plotted in figure 2 for a range of Freon free-stream Mach numbers and is designated as curve (2).

It is now possible to write an approximate expression for c_{nA}/c_{nF} as

$$\frac{c_{n_A}}{c_{n_F}} = \frac{\left(\frac{q_0}{H}\right)_F}{\left(\frac{q_0}{H}\right)_A} \frac{d\left(\frac{p}{H}\right)_A}{d\left(\frac{p}{H}\right)_F} \quad (6)$$

Equation (6) is presented in figure 2 as curve (3), the product of curves (1) and (2). For comparative purposes the normal-force conversion curve of reference 1 has been reproduced in figure 2 as c_{n_A}/c_{n_F} (curve 4). The order of magnitude and trends of the conversion curve of reference 1 are seen to be closely duplicated throughout the Mach number range by the curve derived with the linear approximation. The differences between the two curves, (3) and (4), are largely due to the fact that the analytical curve has been derived with the use of $\frac{d(p/H)}{dM_F}$ equal to its value at the free-stream Mach number as a mean value; whereas, the actual mean value would more generally correspond to some other Mach number close to but probably higher than the free-stream Mach number.

APPARATUS AND PROCEDURES

Tunnel Modification

The design of the temporary modification to the test section of the Langley low-turbulence pressure tunnel to permit testing in Freon-12 at a Mach number of 1.185, corresponding to a Mach number of 1.2 in air, was developed by the method of characteristics. The modification consisted of a plaster nozzle liner built up on the side walls of the tunnel test section entrance cone smoothed and faired with a plastic coating to conform to the design specifications.

The Mach number distribution through the test section, presented as air values converted from the Freon data, is shown in figure 4. The actual free-stream Mach number, as converted from Freon to air, in the region of the model used in these tests is seen to correspond more closely to an average value of 1.205 than to the design value of 1.2. For the comparison purposes employed in the following analysis any such slight difference in the air free-stream Mach number is negligible.

Model and Tests

The wing-fuselage model tested in the Langley low-turbulence pressure tunnel in Freon-12 was the same one previously tested in the

Langley 8-foot high-speed tunnel in air and described in reference 5. The wing had 45° sweepback of the quarter-chord line, an aspect ratio of 4, a taper ratio of 0.6, and NACA 65A006 airfoil sections parallel to the air stream. The fuselage had a fineness ratio of 10 formed by cutting off the rear 1/6 of a body of revolution with a fineness ratio of 12.

The model had been tested in air as a full-span sting-mounted configuration but was tested in Freon as a semispan model mounted with the plane of symmetry of the fuselage flush against the tunnel wall. The left semispan of the original model was selected for the tests and is shown in figure 5. The three chordwise rows of static-pressure orifices located on both wing surfaces at 20, 60, and 95 percent semispan may be seen in the figure.

The Reynolds number for these tests, based on the mean aerodynamic chord of the wing of 6.125 inches, was 2.25×10^6 as compared with 1.93×10^6 in the 8-foot high-speed tunnel tests. This slight difference in Reynolds number in the two wind tunnels would not be expected to have an effect on the comparison of the results obtained for the test model in the presently considered range of angle of attack from 0° to 6° .

The stagnation pressure for the present tests was 12 inches of mercury. The dynamic pressure was, therefore, about 0.4 of the dynamic pressure in the 8-foot high-speed tunnel tests on the same model. Any small spanwise variations in the angle of attack which may have been present due to aeroelastic effects were consequently less pronounced in the present tests.

RESULTS AND DISCUSSION

Pressure Distributions

Pressure coefficients were measured in Freon at a free-stream Mach number of 1.19 which, when converted, corresponds to a free-stream air Mach number of 1.205. The coefficients were measured at 20, 60, and 95 percent of the semispan at angles of attack of 0° , 2° , 4° , and 6° . The magnitude of the differences in pressure coefficients measured on the test model at a representative attitude in the two gases is depicted in figure 6 which is presented as an example. Figure 6 also presents the Freon flow pressure coefficients converted to equivalent air values by the stream-tube area ratio method, applied to the free-stream direction as suggested in reference 1, and shows good agreement of the converted Freon data with the values measured in air. Further comparisons between the converted data and available air data are presented in

figure 7 where the agreement is seen to be consistent. The data of figure 7 typify the results obtained at all conditions of the test. Any small discrepancies between the air and converted Freon data which may be found to exist in the region of steep pressure gradient near the wing leading edge may be attributable to such differences between the two testing conditions as angle-of-attack setting, model surface condition, and aerodynamic twist.

Force and Moment Conversion Factors

The results of the pressure-distribution measurements exemplify the conformity of data obtained in the two testing mediums and justify the use of the conversion method, based on the assumption of similar area ratios in the two gases, at a low supersonic Mach number. The conversion of data obtained in Freon, other than pressure distributions, to air values through the transonic Mach number range, therefore, can be facilitated by following the procedure employed at subsonic Mach numbers in reference 1.

Normal force and pitching moment. - Pressure distributions of reference 6 measured through the transonic Mach number range on the test model in air at angles of attack from 4° to 20° were converted to Freon pressure coefficients and the increments between the load coefficients in the two gases were determined (for example, fig. 8). The percentage increments between the normal-force and pitching-moment coefficients in the two gases were calculated by integration of the load-coefficient increments and thus the corresponding conversion curves of reference 1 were extended to a Freon free-stream Mach number of 1.185 (fig. 9). The data points of figure 9 which are at widest variance with the faired curves were found to represent such small values of c_{nA} and c_{mA} that their conversion to equivalent Freon values, c_{nF} and c_{mF} , would be very little affected by a choice of either the increment percentage suggested by the specific data point or the faired curve. The extended conversion curves of figure 9 are in accordance with the predictions of the linearized approximation of figure 2.

Zero-lift drag. - Wake drags measured in the two gases may be compared by application of the concept of streamline-flow similarity to the region of the wake as in reference 1. The drag coefficient can be expressed as:

$$c_d = \frac{2}{c} \int_{\text{Wake}} \left(\frac{H_1}{H_0} \right)^{\frac{\gamma-1}{\gamma}} \left[\frac{1 - \left(\frac{p_0}{H_1} \right)^{\frac{\gamma-1}{\gamma}}}{1 - \left(\frac{p_0}{H_0} \right)^{\frac{\gamma-1}{\gamma}}} \right]^{1/2} \left\{ 1 - \left[\frac{1 - \left(\frac{p_0}{H_1} \right)^{\frac{\gamma-1}{\gamma}}}{1 - \left(\frac{p_0}{H_0} \right)^{\frac{\gamma-1}{\gamma}}} \right]^{1/2} \right\} dy_1 \quad (7)$$

from reference 7 when the wake is measured sufficiently far downstream for the static pressure to have returned to its value in the free stream.

Figure 10 presents the elemental wake-drag conversion factor calculated by means of equation (7) and with the use of the streamline similarity concept for a free-stream Mach number in air of 1.2 as a function of the local wake total-pressure-loss coefficient, $\left(\frac{H_0 - H_1}{H_0 - p_0} \right)_A$. At sub-

sonic Mach numbers the elemental wake-drag conversion factor was found to vary much more with free-stream Mach number than with the wake total pressure loss. Consequently, the value of $\left(\frac{H_0 - H_1}{H_0 - p_0} \right)_A = 0.12$ which was

found to represent an average total pressure loss in the wake for a wide variety of subsonic drag data was selected in reference 1 to provide the increment percentage of wake-drag coefficient. Although a comparison with experimental data was not available, the same value of $\left(\frac{H_0 - H_1}{H_0 - p_0} \right)_A$ and figure 10 were used to extend the wake-drag conversion curve of figure 11 to a Freon free-stream Mach number of 1.185.

Drag due to lift. - The fact that the concept of induced drag at subsonic speeds, which is associated with the rearward tilting of the lift vector due to the trailing vortices, loses its meaning at supersonic speeds raises some question as to the method to be employed in converting Freon-12 drag data to air data at Mach numbers above 1.0. The subsonic drag conversion method of reference 1 was based on the idea that the measured total drag could be resolved into two components: (a) that part of the drag associated with the general field of flow, that is, the rearward tilting of the lift vector at the wing due to the effect of the trailing vortices and (b) that part of the drag associated with losses of total pressure in the immediate vicinity of the wing. Wake surveys made in the vicinity of the wing would include all of the drag of type (b) but practically none of type (a).

Because of the uncertainty regarding the meaning of induced drag in the aforementioned sense at supersonic speeds, it is more convenient to resolve the total drag at supersonic speeds in a different manner. The total drag may be considered to be composed of a pressure drag and a skin-friction drag with the drag axis taken in the free-stream direction. With the axis taken in this direction all the drag, regardless of its source, will be included. Because the wings considered for use at supersonic speeds are thin and because the leading-edge suction is generally small at transonic and supersonic speeds, it is reasonable to assume that the resultant pressure force on this type of wing is perpendicular to the chord plane and that, for the purpose of determining the Freon to air pressure drag conversion factor, $C_{D_L} \approx C_{Na}$. With the assumption that the resultant pressure force is perpendicular to the chord plane all the drag at $C_N = 0$ can be considered to be skin-friction drag. This skin-friction drag is further assumed to remain constant with changes in angle of attack. This latter component of the total drag would always be completely included in a wake survey at $C_N = 0$. The method of conversion of Freon total drags to air drags at supersonic speeds is then to apply the wake-drag conversion factors (fig. 11) to the zero-lift drag, and the normal-force conversion factors (fig. 9) to the remainder of the drag. Any inaccuracies which may exist in the preceding method of analyzing the total drag in a supersonic flow can result in only negligible errors in the conversion of Freon total drags to air total drags inasmuch as the normal-force conversion factors and the wake-drag conversion factors differ by a maximum of only a few percent over the Mach number range considered.

The preceding assumptions on which the supersonic drag conversion factors depend are equally applicable at subsonic speeds for thin wings when the leading-edge suction is small. It is desirable, therefore, to compare the results of both conversion methods at subsonic speeds. A quantity of available Freon total drag data, obtained at high subsonic speeds, were converted to corresponding data in air by the method of reference 1 and also by the method of the present report. The converted total drag data obtained by both methods differed by a maximum of less than two percent. Inasmuch as the drag-conversion method of reference 1 is more generally applicable at subsonic speeds than the conversion method of this report, subsonic drag data obtained in Freon in the low-turbulence pressure tunnel will continue to be converted to air data by the earlier method.

Lateral force and moments. - The moment of a wing about a longitudinal axis is a function only of the normal force acting on the wing for a constant center of pressure. The spanwise section-normal-force coefficients over the wing for any given free-stream Mach number are all converted by the same percentage increment (fig. 9) so that the center of pressure location is the same for both air and Freon-12. The conversion factor

for the normal-force coefficient, therefore, can be used to convert rolling-moment coefficients measured in Freon-12 to corresponding coefficients in air.

The yawing-moment and side-force conversion factors will depend upon the nature of the forces involved and so may vary for different configurations. For example, the yawing moment contributed by a deflected rudder is caused by a pressure force acting on the vertical tail surface and the conversion of the yawing-moment coefficient from Freon-12 to air should be made, therefore, by application of the normal-force conversion factor. On the other hand, a yawing moment may be caused by asymmetrical skin-friction forces, in which case, the moment should be converted from Freon-12 to air by application of the wake-drag conversion factor. The conversion of yawing moment and side force, therefore, should be determined for each individual configuration after consideration of the forces involved.

CONCLUDING REMARKS

A temporary modification was made to the test section of the Langley low-turbulence pressure tunnel to permit testing in Freon-12 at a low supersonic Mach number. A comparison between the force data obtained on a wing in air and in Freon-12 at a low supersonic Mach number has been obtained from the pressure coefficients measured at three spanwise stations on a 45° sweptback wing at angles of attack of 0°, 2°, 4°, and 6°. The pressure-coefficient data of the present Freon tests, as converted to equivalent air values, corresponded very well to measured pressure-coefficient data obtained in air.

Conversion factors to be applied to force and moment data obtained through the transonic Mach number range in Freon-12 for reduction to equivalent air values are presented as computed on the assumption of streamline similarity of the flow in both Freon-12 and air.

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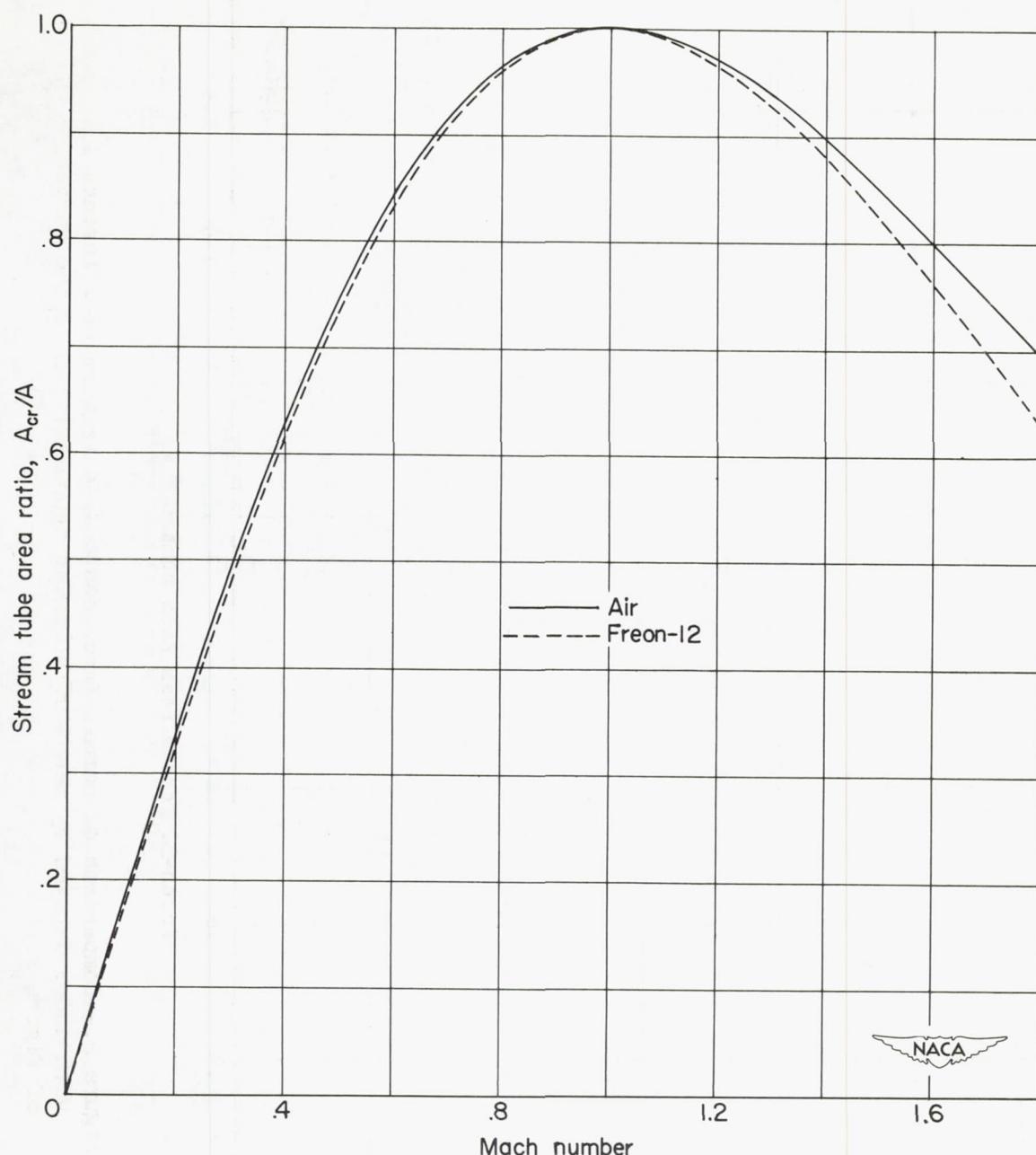


Figure 1.- Variation of stream-tube area ratio with Mach number.

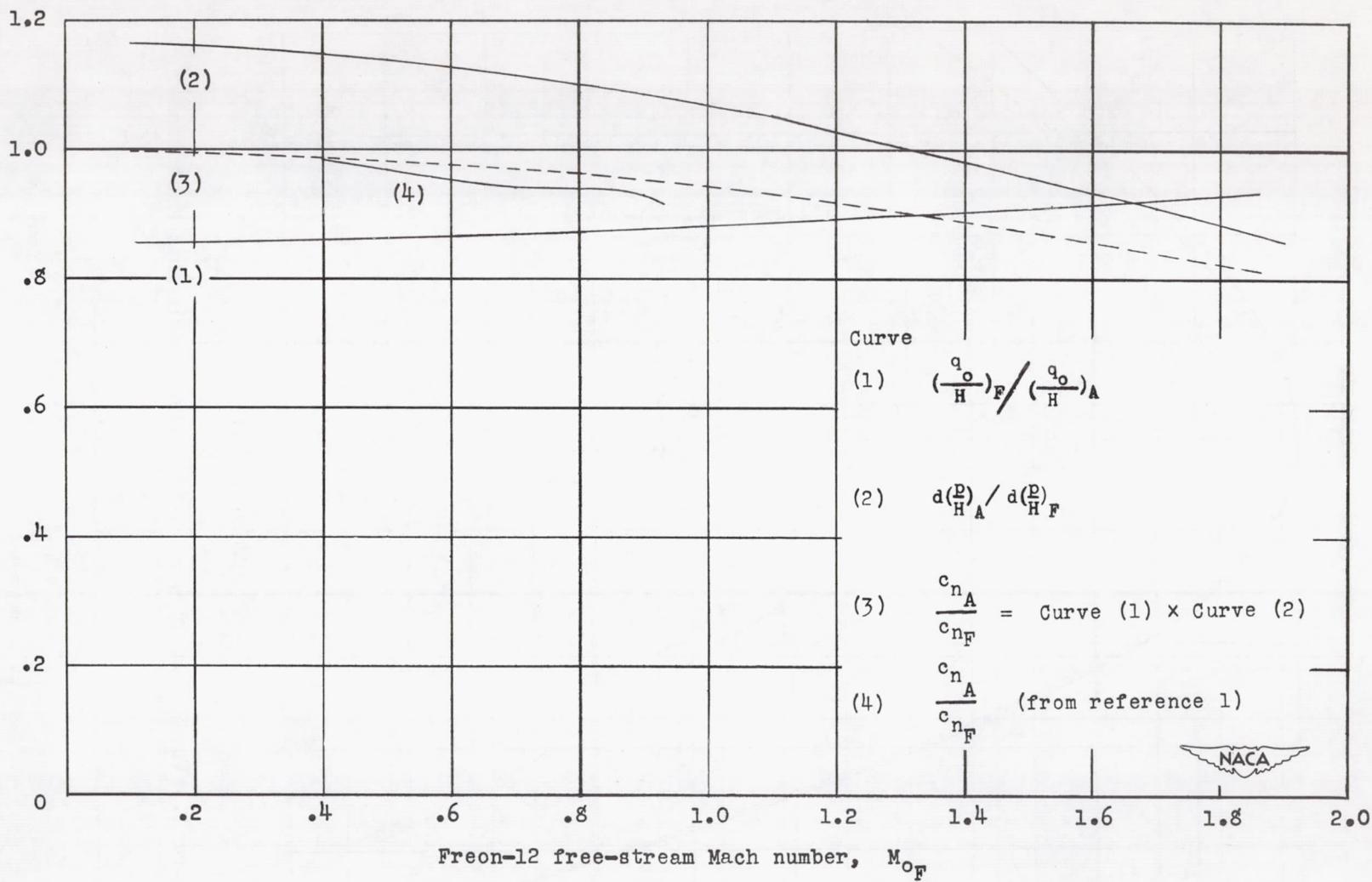


Figure 2.- Comparison of normal-force conversion factors of reference 1 with those derived by means of a linear approximation to the curves of figure 3.

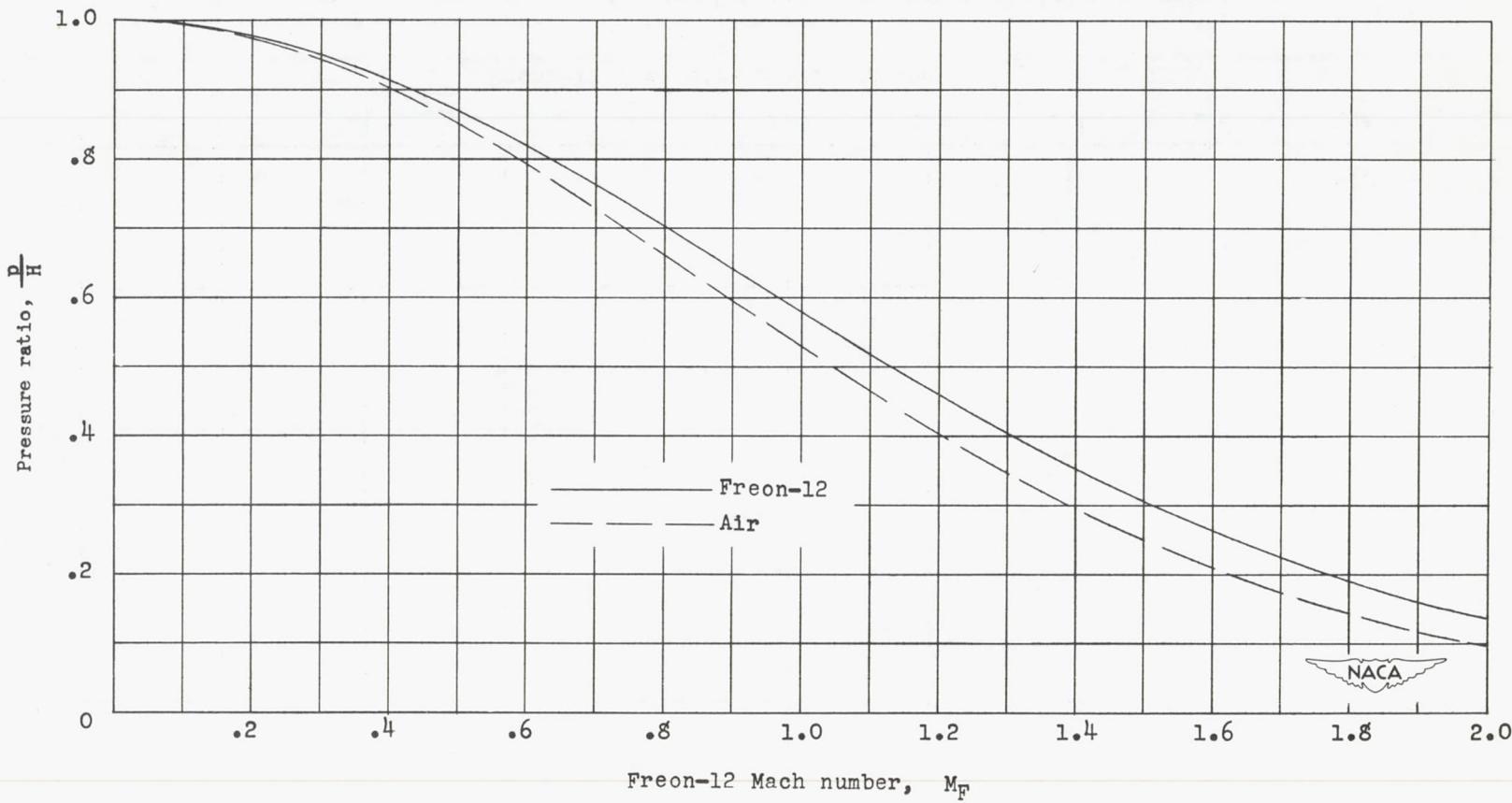


Figure 3.- Variation of pressure ratio for Freon-12 and air with Freon-12 Mach number. The pressure ratios for air are determined for similar area ratios in the gases.

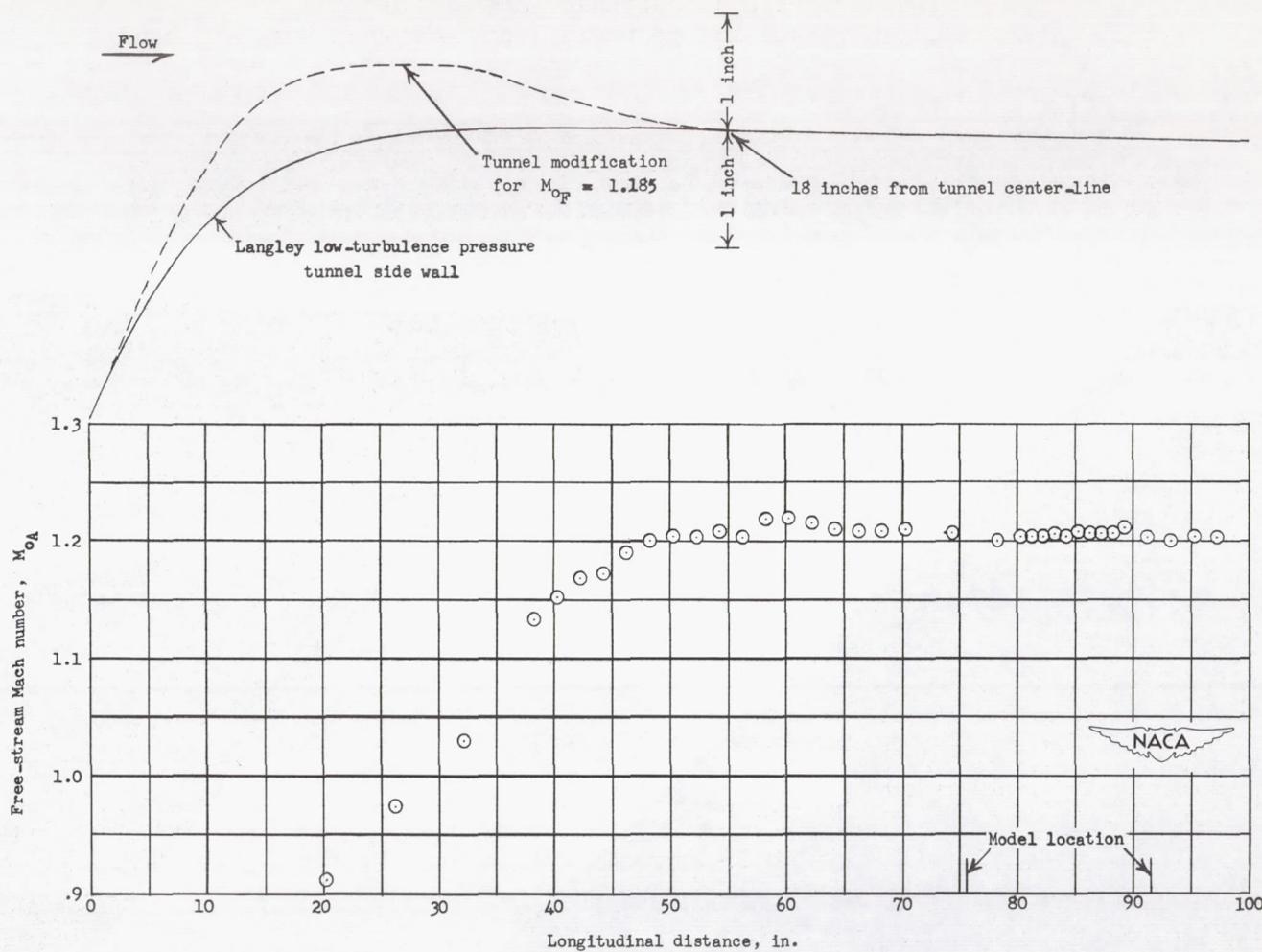


Figure 4.- Longitudinal distribution of Mach number midway between the center line and the side wall of the Langley low-turbulence pressure tunnel with the plaster liner installed and Freon-12 as the test medium; Freon-12 data converted to air values.

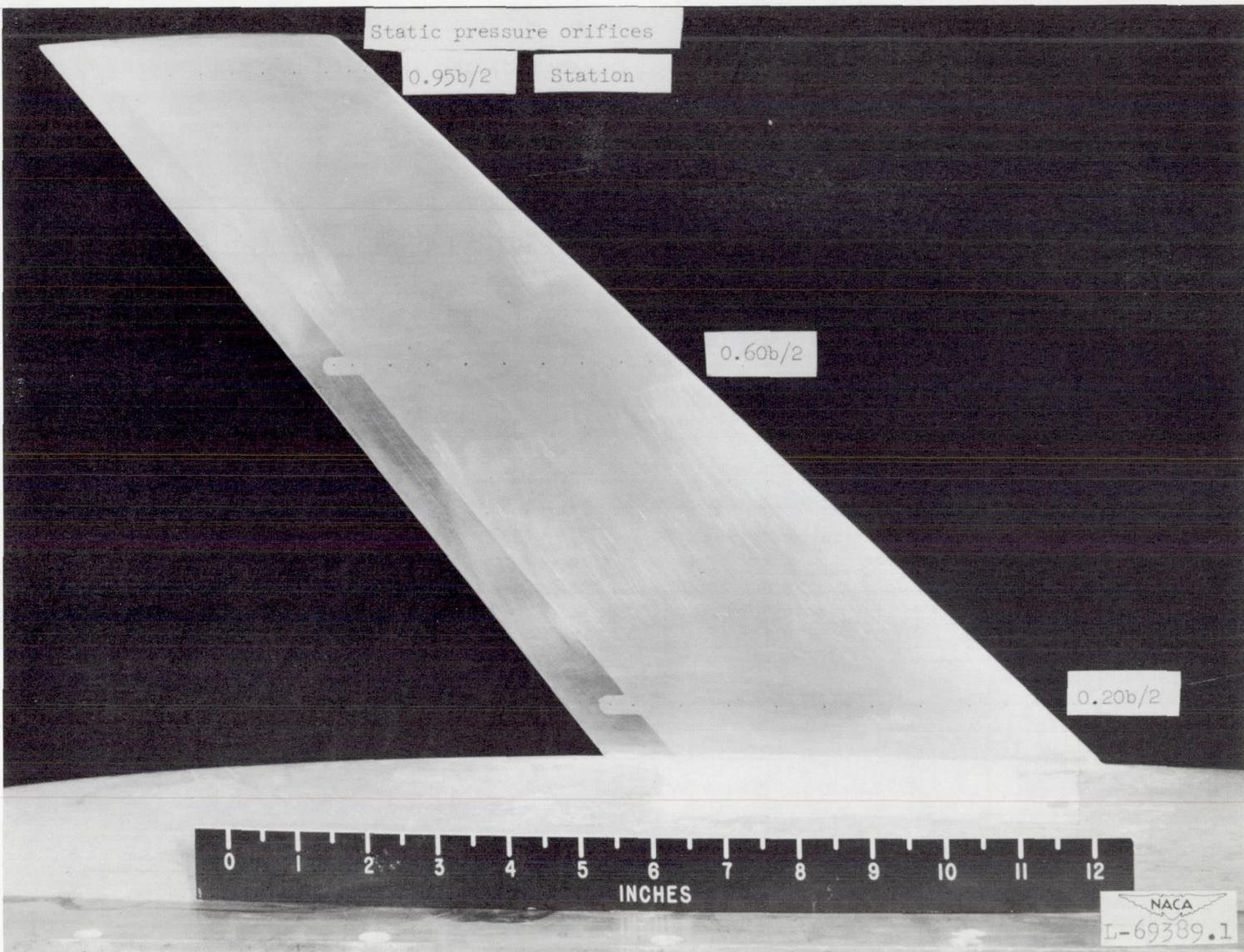


Figure 5.- Wing-fuselage model prepared for installation in test section of Langley low-turbulence pressure tunnel.

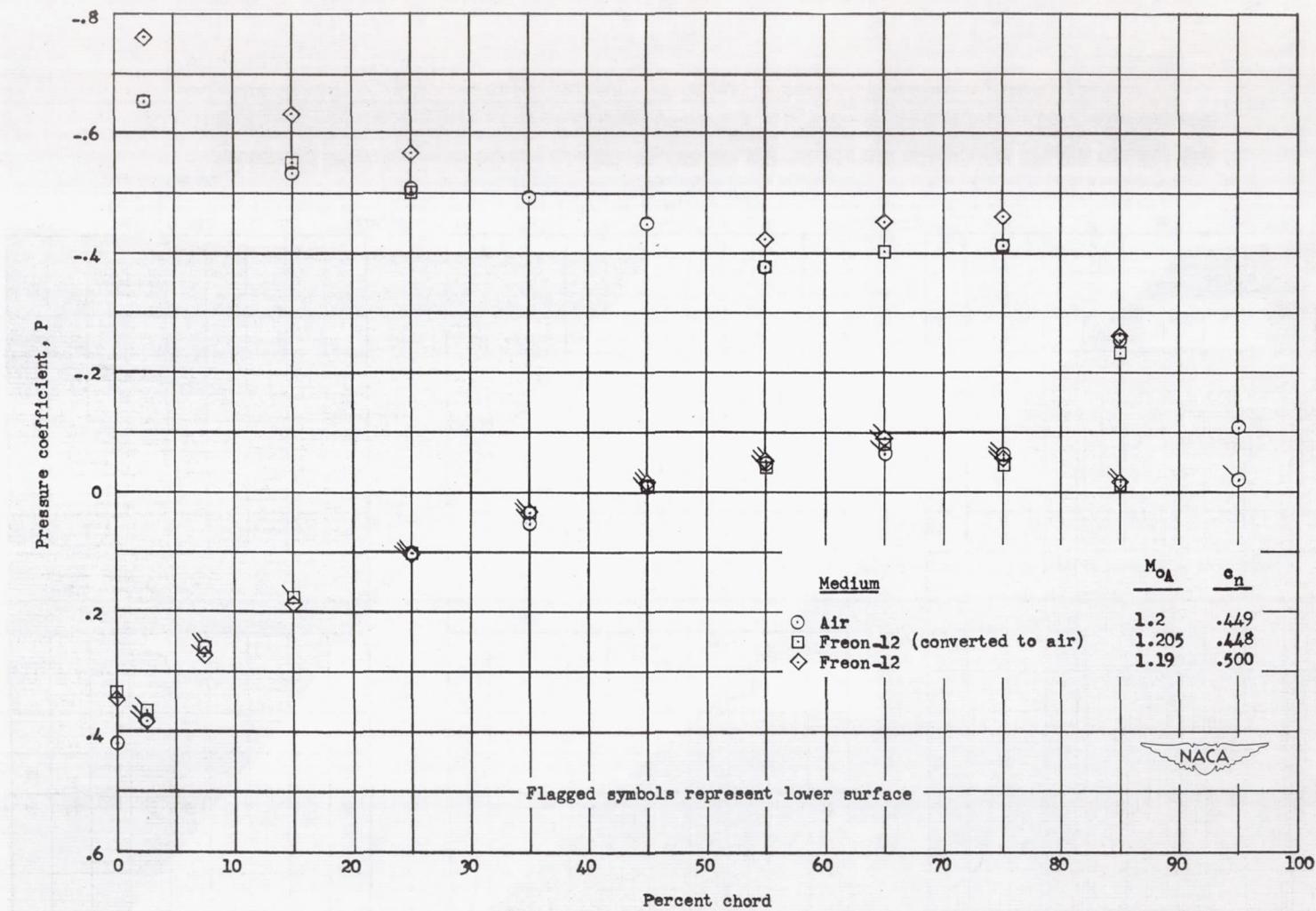


Figure 6.- Comparison of pressure distributions measured at $0.60b/2$ in air and Freon-12 and also converted from Freon-12 to air; $\alpha = 6^\circ$.

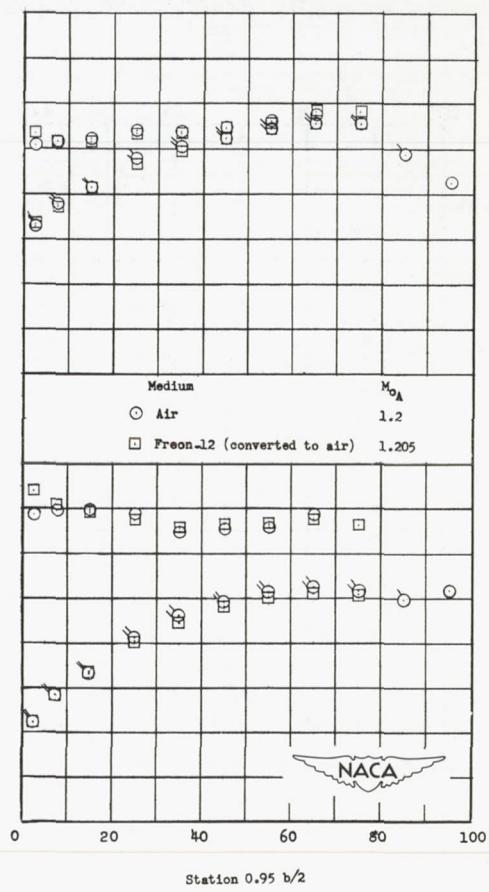
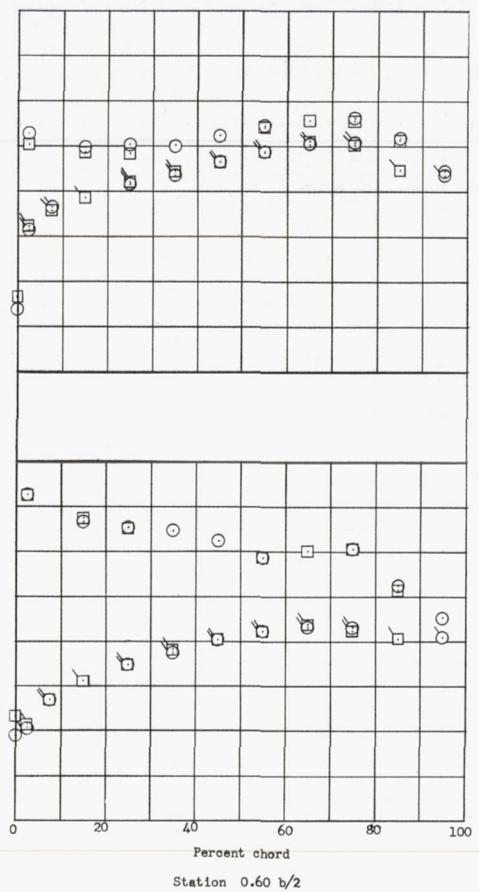
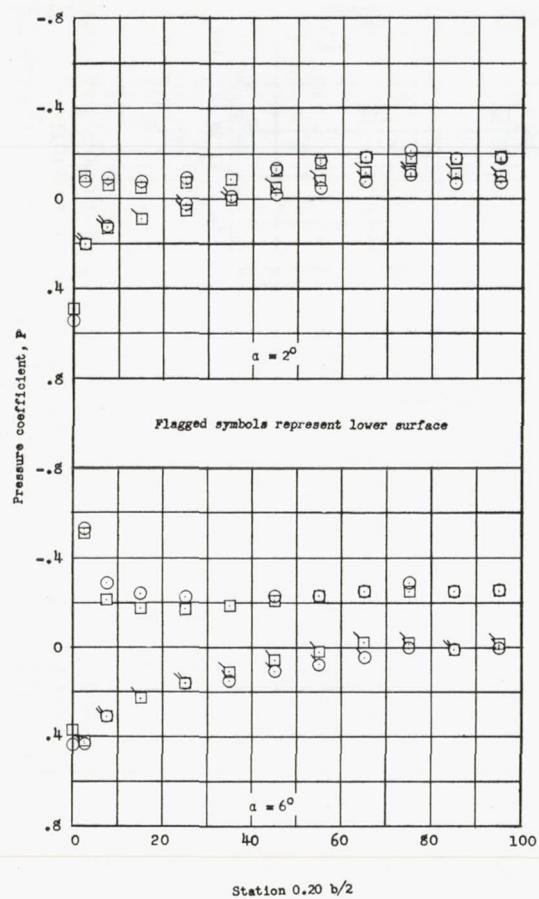


Figure 7.- Chordwise pressure distributions at three semispan stations of the NACA 65A006 wing-fuselage model. Freon-12 data converted to equivalent air values.

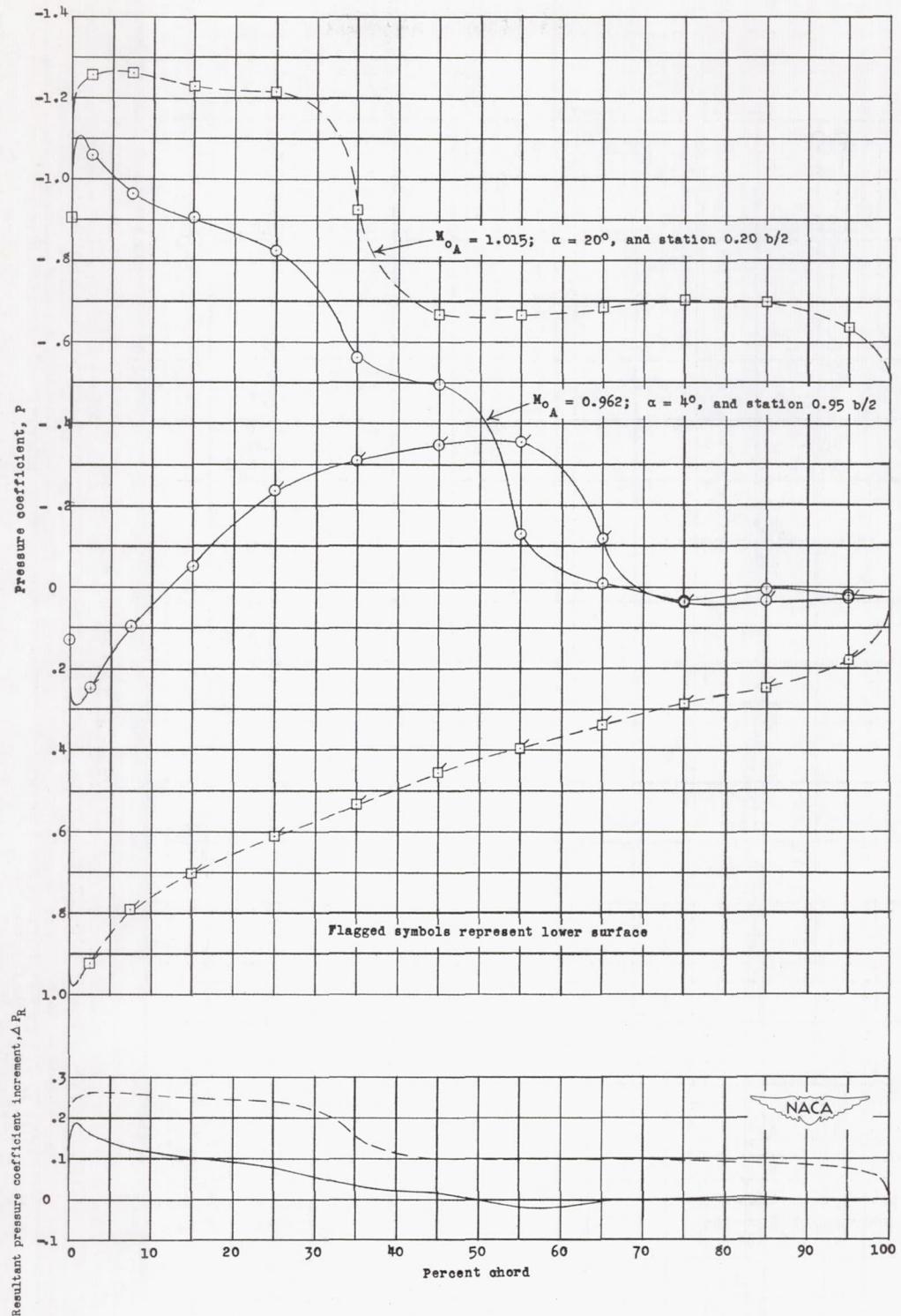


Figure 8.- Chordwise variation of upper- and lower-surface pressure coefficients in air and the calculated resultant-pressure-coefficient increments required to convert from air to Freon-12 for the NACA 65A006 wing-fuselage combination.

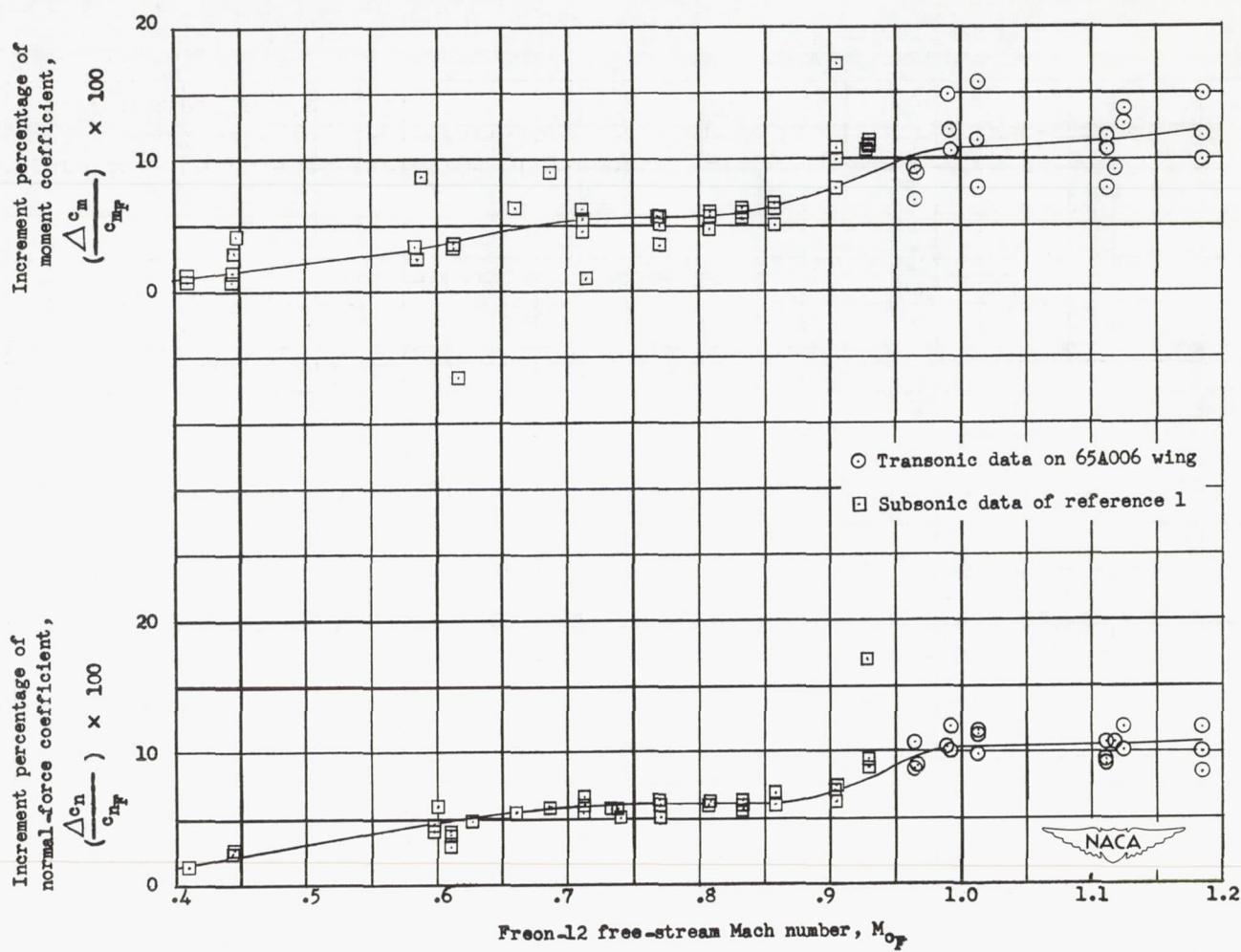
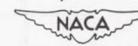
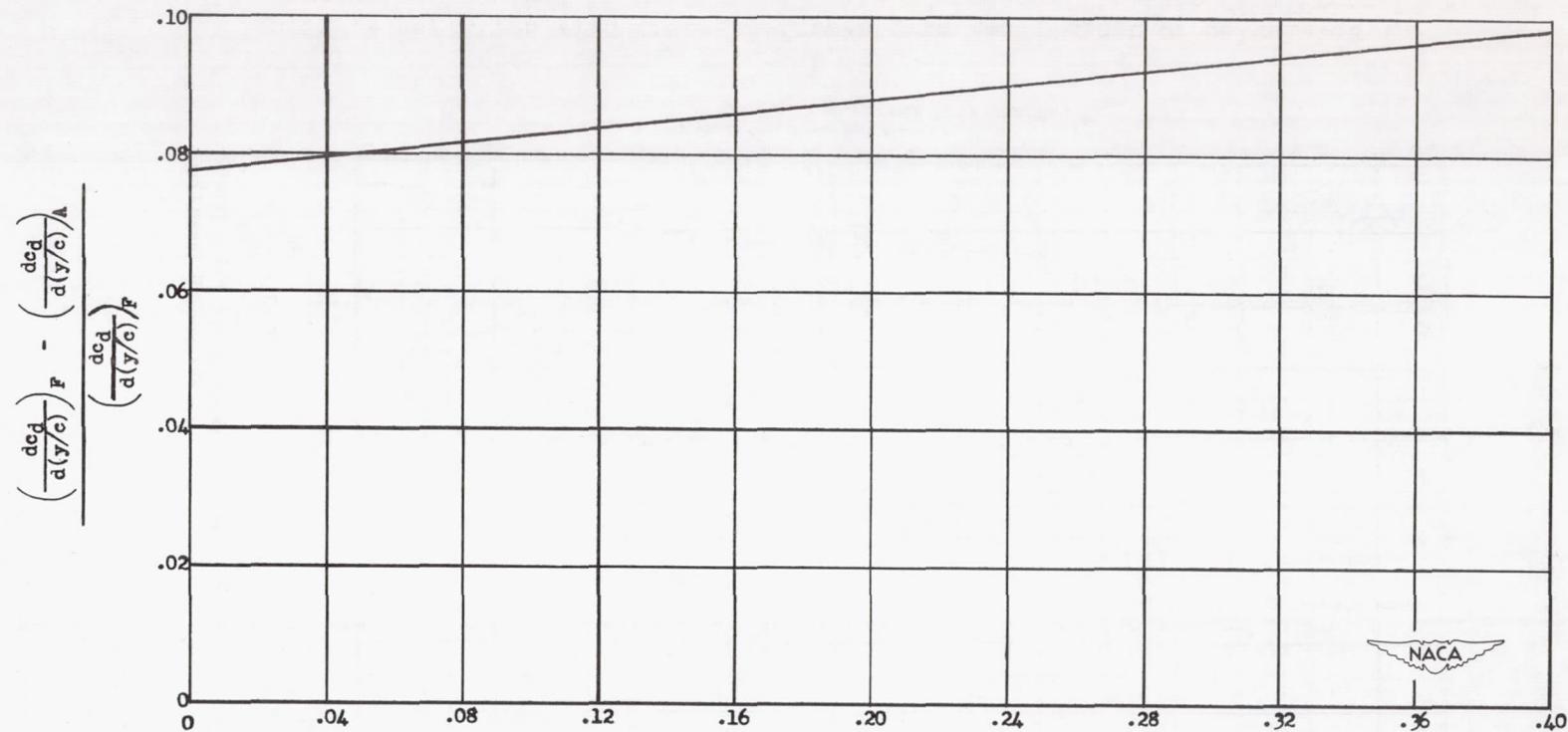


Figure 9.- Variation with Freon-12 free-stream Mach number of calculated increments required to convert lift and moment coefficients from Freon-12 to air.

Elemental wake-drag conversion factor,



Local wake total-pressure-loss coefficient in air, $\left(\frac{H_o - H_1}{H_e - p_o} \right)_A$

Figure 10.- Calculated variation of elemental wake-drag conversion factor with local wake total-pressure-loss coefficient in air for a free-stream Mach number in air of 1.2.

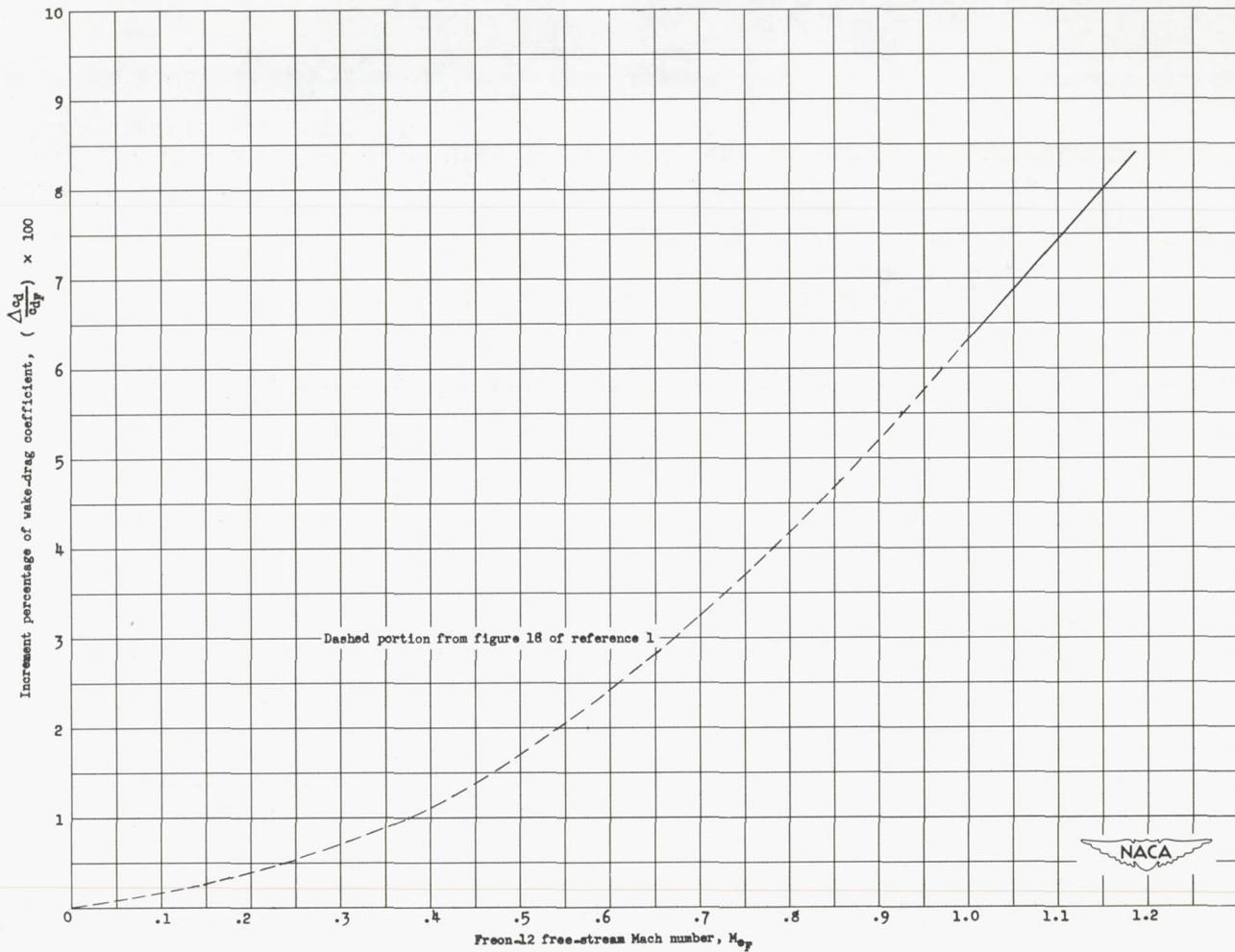


Figure 11.- Variation with Freon-12 free-stream Mach number of calculated increments required to convert wake drag coefficient from Freon-12 to air.